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RESEARCH MEMORANDUM

FACTORS AFFECTING TRANSITION AT SUPERSONIC SPEEDS

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With the advent of flight at supersonic speeds there has been renewal of interest in the subject of boundary-layer transition. Whereas experience has shown that extensive runs of laminar flow cannot be obtained under practical field operating conditions at subsonic speeds, both theory and practical considerations indicate a more favorable outlook at supersonic speeds. For example, it has been demonstrated that longer runs of laminar flow can be obtained by cooling the boundary layer and that the cooling can be obtained by taking advantage of the natural heat capacity of a missile, at least in the initial phases of the flight. Also, since the missile is intended to make but a single flight, the construction and maintenance of a smooth surface is simplified. Further, such large reductions in drag and aerodynamic-heating rate are possible with laminar flow that reexamination of the problem of transition is imperative. This paper surveys the available material to summarize what is known to date about boundary-layer transition at supersonic speeds.

The bulk of our current information on supersonic transition comes from wind tunnels. As in subsonic tunnels, the transition results obtained are critically dependent on the quality of the airstream. It is necessary, therefore, in any analysis of tunnel transition data to first ascertain whether the results are unduly affected by wind-tunnel disturbances. Indications have been found that supersonic transition data are affected by local shocks and angularity of the tunnel airstream as well as by turbulence level. Because it is difficult to evaluate the quality of supersonic tunnel flows by direct measurement of these factors, the NACA is conducting comparative transition tests with zero heat transfer on a particular body shape, a 10° cone, in many of its supersonic facilities. In figure 1 are shown some of the results obtained to date. The Reynolds number of transition R_t, based on distance from the nose, is plotted against M and also against R per foot. Both abscissas are used here simply to define the test conditions and not to indicate that they are significant parameters affecting transition.

This figure is presented only to show the wide range of transition Reynolds numbers obtained in different tunnels under comparable test conditions and hence the wide variation in the quality of the airstreams in these wind tunnels. Some of the facilities have sufficiently small

disturbances to permit extensive laminar flows, for example, the Langley 9-inch and 4-foot supersonic tunnels.

In the remainder of this paper the bulk of the tunnel data used are from these two tunnels having the high transition Reynolds numbers. In addition, transition data from model flight tests in still air at the U.S. Naval Ordnance Laboratory and at the Ames Aeronautical Laboratory are used.

The effect of Mach number on transition on smooth bodies at supersonic speeds is considered in figure 2. The data presented at the lower Mach numbers, $M = 5$ or less, are for zero or essentially zero heat transfer. The data at the higher Mach numbers include some boundary-layer cooling. The point at $M = 0$ is the transition Reynolds number for a flat plate at low speeds for a wind-tunnel turbulence level of less than 0.1 percent (ref. 1). For the lower Mach number tests, R_t generally corresponds to transition at the model base; hence there are no changes in pressure gradient to be considered. The arrow at $M = 5.8$ (data from ref. 2), incidentally, indicates that the exact value of R_t is not known but is greater than the value plotted.

In general, the results in figure 2 for M less than 5 indicate a decrease in R_t with increasing Mach number except for the cone-cylinder when M is less than 2. It may be remarked here that the rate of decrease in transition Reynolds number with increase in Mach number may be affected somewhat by changes in tunnel-flow characteristics that occur with changes in test section Mach number. From these data one might expect to obtain very little laminar flow at higher Mach numbers and this was the picture until recently. Recent hypersonic wind-tunnel results, however, show the relatively high values of R_t indicated by the points for $M \approx 6$ and 7. These relatively high values of R_t are believed to be due partly to favorable heat-transfer effects which may usually be expected at hypersonic speeds and partly to favorable shock—boundary-layer interactions at the nose of the models which result in a favorable local pressure gradient (ref. 3). The important conclusion that can be drawn is that values of R_t of the same order of magnitude as those obtained at low supersonic speeds can be obtained in practical cases at hypersonic speeds.

Figure 3 shows the effect of surface pressure gradient on smooth bodies at a Mach number of 1.61. The sketches in the upper part of the figure indicate the types of bodies tested and their pressure distributions. The curves in the lower part of the figure are a plot of the measured skin friction based on wetted-surface area. At the point where the experimental skin-friction curve leaves the theoretical laminar curve, transition has appeared at the base of the body and is beginning to move forward.

The results indicate that the parabolic body with a moderately favorable pressure gradient over the length of the body had the largest value of R_t, about 11×10^6. The cone-cylinder with the least amount of favorable pressure gradient showed the lowest value, about 2.75×10^6. From these results, it is apparent that pressure gradient has a strong effect on transition at the lower supersonic Mach numbers just as at subsonic speeds. In order to obtain high values of R_t, it is apparently desirable to maintain a favorable pressure gradient where the boundary layer is most susceptible to instability - in these tests a favorable pressure gradient toward the rear of the body. At higher test Reynolds numbers, when transition has moved forward on the bodies, both the ogive-cylinder and cone-cylinder show larger runs of laminar flow than the parabolic body because of the more favorable pressure gradients on the ogive or at the cone shoulder.

Some additional results showing the effects of pressure gradient are presented in figure 4. In this case the pressure gradient was altered by changing the shape of the body progressively from that shown at the upper left to that at the upper right. The transition results are plotted against the ratio of base area to maximum cross-sectional area, which is a rough index of the increase in length of favorable pressure gradient. It may be noted that increasing the run of favorable pressure gradient resulted in a reduction in the rate of falling pressure. Transition in these tests always occurred at the base.

The results indicate a large increase in R_t with increase in length of favorable pressure gradient at both Mach numbers investigated. The reverse in the curves at the lowest area ratio is due to laminar separation at the model base. The reason for the discontinuity in the Mach number 1.93 curve near $A_{base}/A_{max} = 0.7$ is not known.

An analysis of the data from which the curves of figures 3 and 4 were obtained and of other results available at supersonic speeds shows a tendency for the favorable effects of a falling pressure to decrease as the boundary layer becomes thin as near the nose of a body or at very high test Reynolds numbers. In addition, theoretical calculations by Lees (ref. 4) and by Weil (ref. 5) predict a decrease in the effects of pressure gradient as M is increased; although, as yet, there is no reliable experimental verification.

The possibility of a large stabilizing effect due to cooling of the laminar boundary layer at supersonic speeds in the case of the Tollmien-Schlichting type of boundary-layer instability was predicted theoretically in the well-known work of Lees in 1947 (ref. 6). Recent studies, particularly those in the Langley 4- by 4-foot supersonic pressure tunnel

(refs. 7 and 8) and in flight (ref. 9), have confirmed the existence of this effect. In figure 5, the chart on the right compares the theoretical effect of heat transfer on the stability of the boundary layer on a flat plate (ref. 10) with the experimental effect of heat transfer on transition on the RM-10 parabolic body. The parameter R_t is plotted against T_w/T_∞, the ratio of wall temperature to free-stream temperature. Regions to the left of the curves indicate either a theoretically stable or experimentally laminar boundary layer. At a value of this ratio of 1.05 theory indicates that the boundary layer will be stable for all Reynolds numbers. The trends of the curves are in good agreement. A part of the displacement between curves occurs because of the comparison between two- and three-dimensional bodies, a part because of the additional length of surface required for the disturbance in the boundary layer to amplify sufficiently to break down the laminar flow, and another part because of the favorable pressure gradient on the body. The highest value of R_t obtained in the tunnel tests was about 28.5×10^6 (ref. 8). The highest value of R_t measured to date with cooling is about 90×10^6 and was obtained at White Sands Proving Ground in flight on the conical nose of a V-2 rocket (ref. 9). Thus, if transition can be limited to the apparently Tollmien-Schlichting type, boundary-layer cooling will be of great aid in obtaining long runs of laminar flow.

In the chart on the left the experimental results for the parabolic body have been replotted against $\Delta T/T_{stag}$, an index of the amount of heating or cooling relative to the stagnation temperature. In addition are shown some results typical of the earlier experiments in other wind tunnels in which low adiabatic transition Reynolds numbers were obtained.

An analysis of the results shows that when the transition Reynolds number for zero heat transfer is low, the effects of heat transfer are small, and, when R_t for the adiabatic case is high, the effects of heat transfer are large. The low effectiveness of heat transfer on transition in the earlier tests is usually derived from the fact that transition is generally influenced by surface roughness, boundary-layer separation due to adverse pressure gradients, or tunnel effects. These types of transition do not appear to be strongly influenced by heat transfer.

Because of its importance, the next type of transition to be studied is that due to surface roughness. In figure 6 is presented a plot of $R_t/R_{t_{k=0}}$, the ratio of Reynolds number of transition with single-element surface roughness to Reynolds number of transition for a smooth body, against the parameter k/δ^*_k, the ratio of roughness height to boundary-layer displacement thickness at the roughness. The solid line is the low-speed correlation obtained by Dryden (ref. 1) on the basis of transition data for Reynolds numbers less than 2×10^6. For this case, the

results show that for a roughness-height ratio of less than 0.1 single-element surface roughness has no effect on transition. Results for bodies having values of R_t greater than 2×10^6 do not extend to sufficiently low values of k/δ^*_k to establish the validity of this conclusion for cases with longer runs of laminar flow.

Only one approximate point is available for plotting for the supersonic speeds. This point indicates a somewhat higher value of roughness ratio required to effect transition than in the subsonic case, but the point may be within the range of scatter obtained in the subsonic correlation. A somewhat larger amount of data is available for comparison with subsonic results if the Reynolds number for transition itself is plotted against roughness ratio as is indicated by the chart on the left in figure 7. The three data points for the parabolic body at $M = 1.61$ appear to fall within the same range as the low-speed airfoil data for similar single-element roughness. The steep rise in R_t as the roughness ratio is reduced in the supersonic case compares closely to the trends obtained at high Reynolds numbers of transition subsonically.

In the chart on the right is presented a plot of R_t against the parameter $\frac{k}{\delta_{R=10^6}}$ for distributed surface roughness on an ogive-cylinder body. When the roughness is distributed over an area it is not clear what value of boundary-layer thickness should be used as an index of the roughness effect; hence, an arbitrary value of boundary-layer thickness, δ for $R = 10^6$, was chosen for this chart. The tests were made with a wall-to-free-stream temperature ratio of about 1.04, thus indicating that the tests were within the region for infinite Tollmien-Schlichting boundary-layer stability for a flat plate. The results show trends similar to those determined for single-element roughness. Other preliminary data indicate that, for equivalent roughness heights, transition will occur at lower Reynolds numbers for distributed roughness than for single-element roughness when the leading edges of the roughnesses are at the same location.

An investigation of effects of heat transfer on transition due to roughness was made on the parabolic body at $M = 1.61$ (refs. 7 and 8) but few of the data were susceptible to the present type of analysis. A study of the trends, however, shows that the effect of heat transfer on the critical roughness parameter may be small. In particular, however, the results showed that whenever transition was significantly affected by surface roughness or, for that matter, by any other type of finite disturbance, then boundary-layer cooling was ineffective in extending the length of the laminar run.

If the results that have been presented on surface roughness are interpreted to mean that the Mach number effects on the correlations

are small, then for constant Reynolds number the allowable roughness height before transition is effected should increase with Mach number because of the growth of boundary-layer thickness with Mach number. At $M = 5$ the allowable roughness should be increased by 2.5 and at $M = 10$ by a factor of 6 (fig. 8, left plot).

In addition, as the altitude is increased or the pressure decreased, the molecular mean free path becomes relatively large compared to the protuberance height and continuum flow will not exist and the effects of surface roughness may conceivably disappear. Calculations indicate that for all cases where surface roughness effects could be detected the roughness was considerably greater than 100 times the length of the molecular mean free path (fig. 8, right plot). The calculations also show that, even on the basis of this criterion, the allowable surface roughness will be greater than 200 microinches at 100,000 feet and 7000 microinches, or 0.007 inch, at 200,000 feet altitude. The shaded area in figure 8 indicates the usual range of maximum surface roughness encountered on wind-tunnel and flight-test models.

Up to now all data that have been presented have been for bodies only and for zero angle of attack. Airplanes and missiles, however, usually have wings and fly at some angle of attack. There are insufficient data on wing transition to present any type of correlation; hence, this phase will not be discussed. Figure 9, however, has been prepared to show the effect of α on R_t for two bodies, each at a different Mach number. The tests of the parabolic body were made in a wind tunnel without heat transfer and transition was obtained from force tests and boundary-layer surveys. The results thus correspond to transition at the base of the body. The tests of the slender ogive-cylinder were made in the Ames free-flight tunnel and include a large amount of cooling. In this case transition was obtained by means of shadowgraph studies and is shown for the upper surface only since this is the more critical surface. The latter tests were also limited to a Reynolds number of 11×10^6.

Both sets of data, which include differences in Mach number and heat-transfer conditions, indicate similar trends: a decrease in R_t as α is increased. For the parabolic body, a change in α from 0° to 2° reduces R_t by 60 percent. Both curves are not too well defined for α less than 1°, but the trends appear to indicate that transition will be sensitive to α even at very low angles.

In conclusion, first, boundary-layer transition should be of the Tollmien-Schlichting type if favorable effects of pressure gradient and heat transfer are to be realized. Maximum transition Reynolds numbers of about 28×10^6 in wind-tunnel tests of a parabolic body and 90×10^6

in flight tests of a cone have been obtained with boundary-layer cooling. The effects of surface roughness at supersonic speeds appear similar to those at subsonic speeds, and the allowable-roughness-height parameters are of about the same magnitude as at subsonic speeds. Hence, to avoid transition due to roughness, the roughness size should be limited to about 1/10 the boundary-layer displacement thickness. Finally, for the longest possible runs of laminar flow, the body should be closely alined with the flow.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., September 1, 1953.

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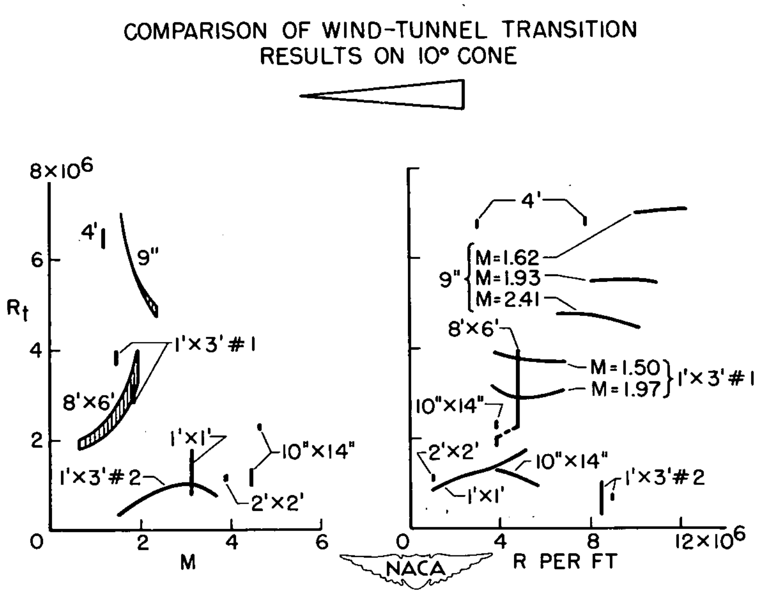

Figure 1

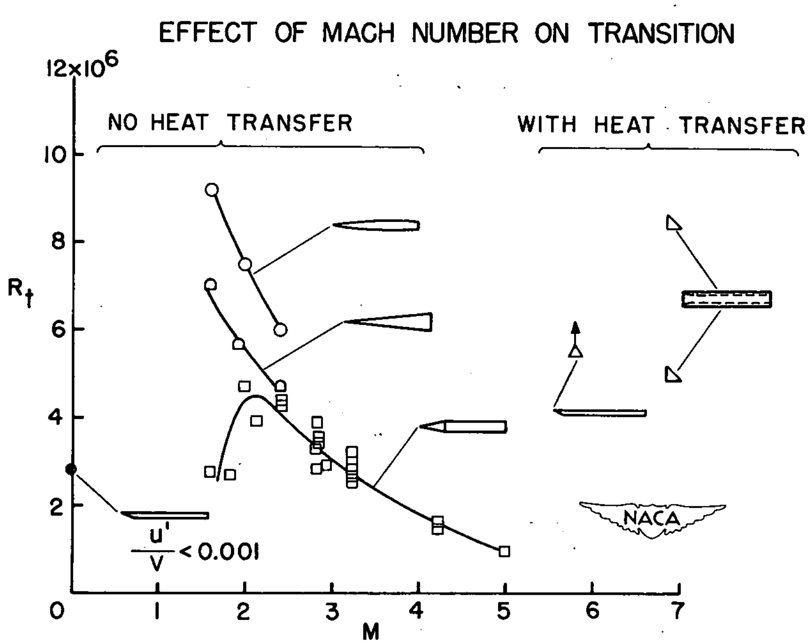

Figure 2

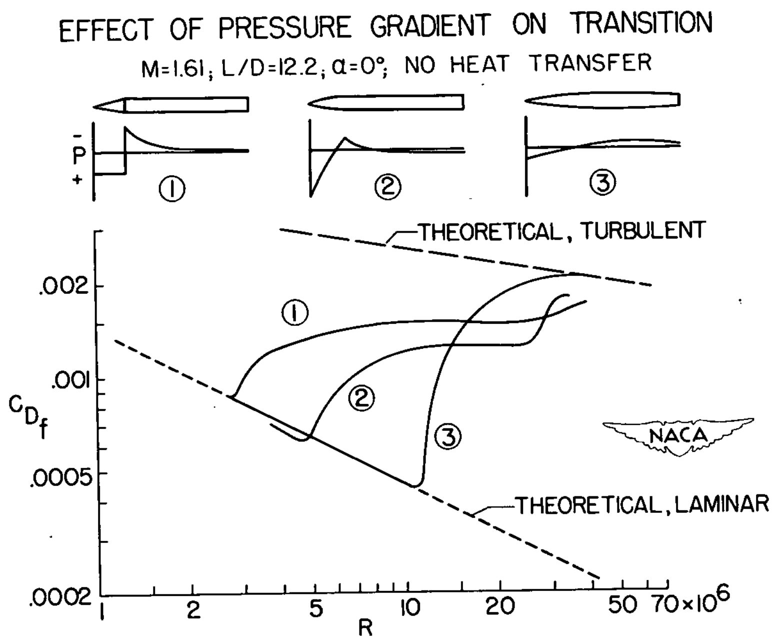

Figure 3

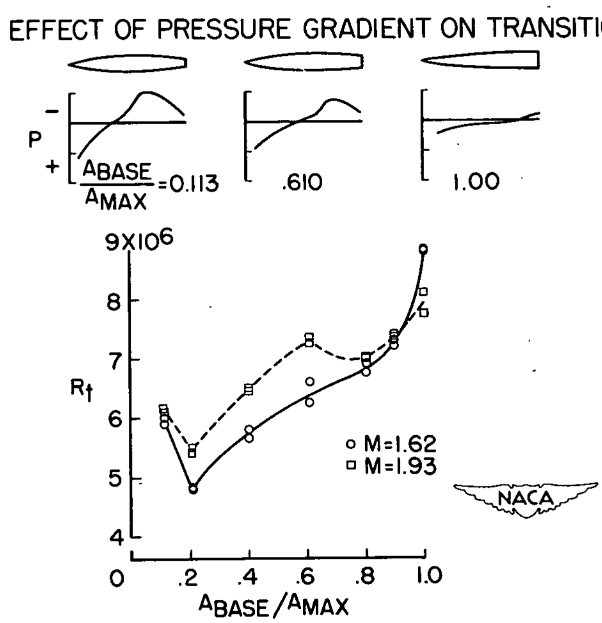

Figure 4

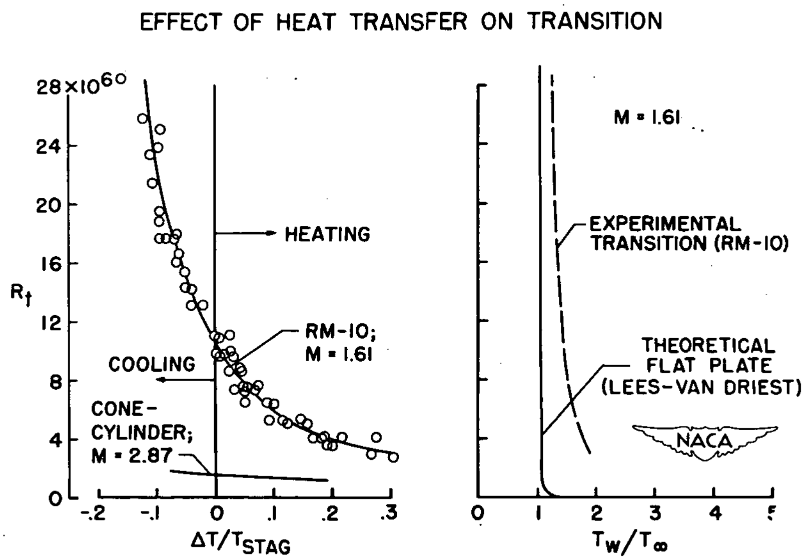

Figure 5

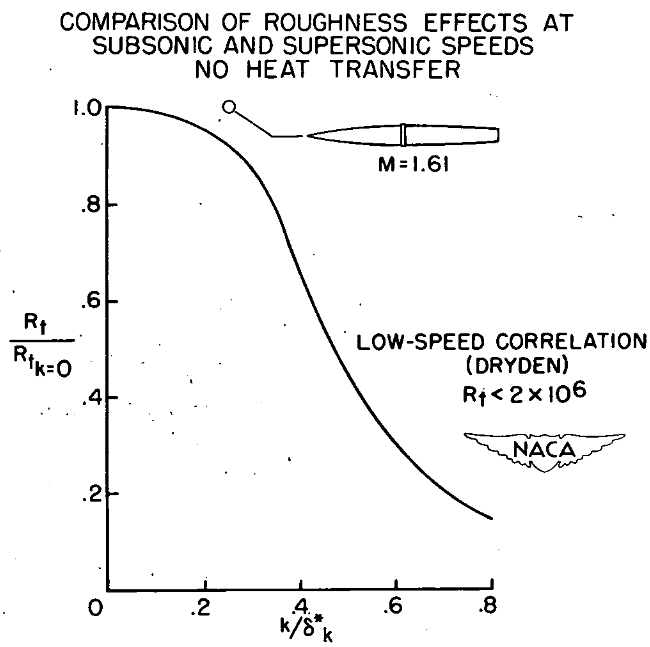

Figure 6

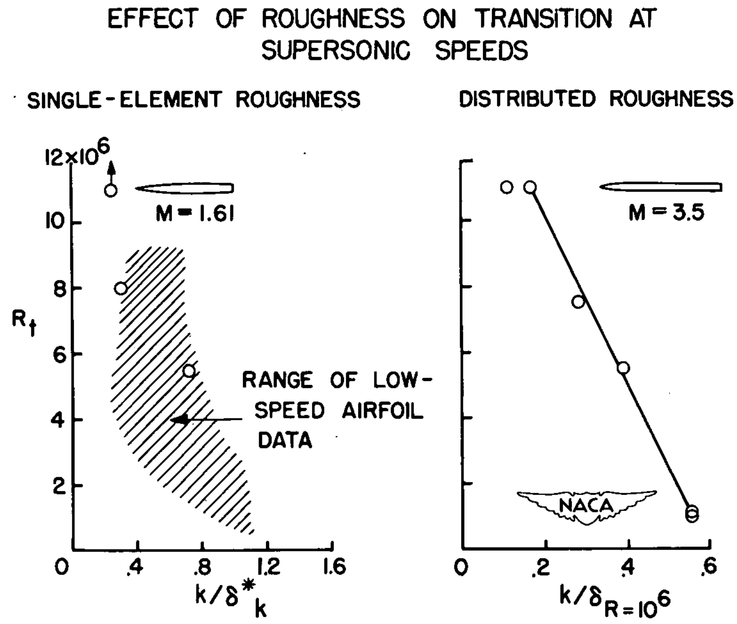

Figure 7

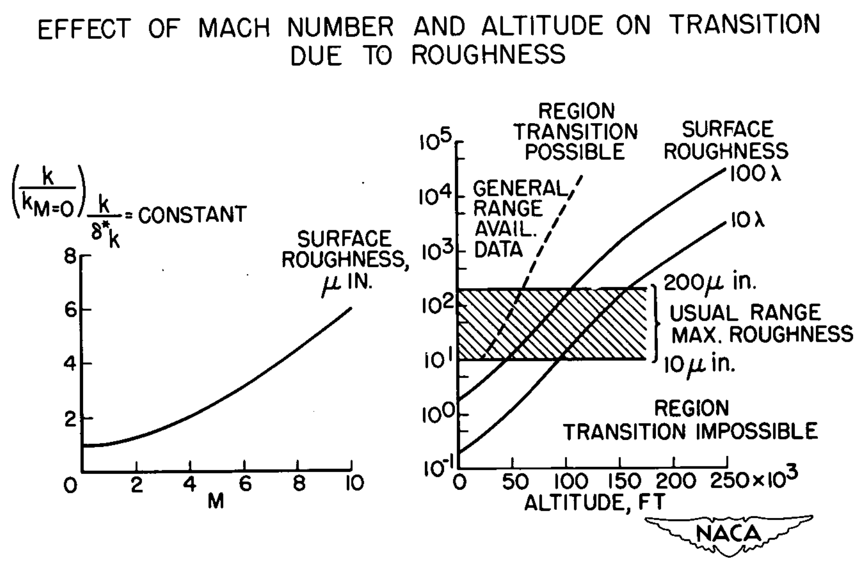

Figure 8

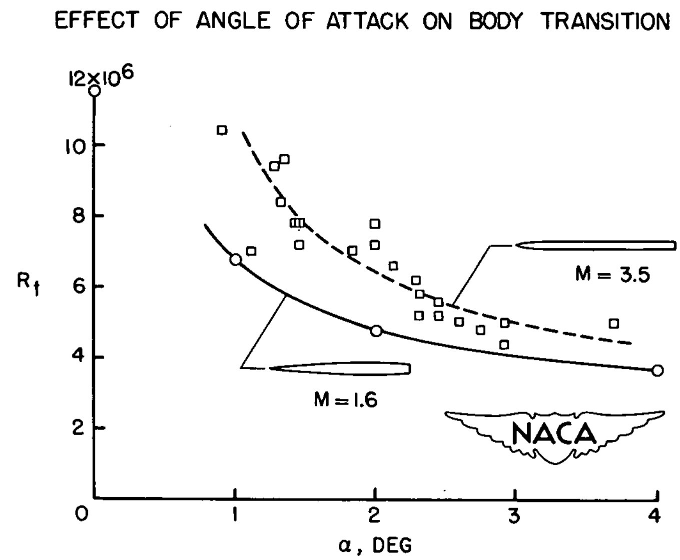
EFFECT OF ANGLE OF ATTACK ON BODY TRANSITION
12x10^6
10
8
6
4
2
0
R_t
M = 3.5
M = 1.6
NACA
1
2
3
4
α, DEG

Figure 9